Solar System Exploration Augmented by In-Situ Resource Utilization: Human Mercury and Saturn Exploration

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Abstract

Human and robotic missions to Mercury and Saturn are presented and analyzed. Unique elements of the local planetary environments are discussed and included in the analyses and assessments. Historical studies of space exploration, in-situ resource utilization (ISRU), and industrialization all point to the vastness of natural resources in the solar system. Advanced propulsion benefitted from these resources in many ways. While advanced propulsion systems were proposed in these historical studies, further investigation of nuclear options using high power nuclear thermal and nuclear pulse propulsion as well as advanced chemical propulsion can significantly enhance these scenarios. Updated analyses based on these historical visions will be presented. Nuclear thermal propulsion and ISRU enhanced chemical propulsion landers are assessed for Mercury missions. At Saturn, nuclear pulse propulsion with alternate propellant feed systems and Titan exploration with chemical propulsion options are discussed.

Nomenclature

3He	helium 3
4He	helium (or helium 4)
AMOSS	atmospheric mining in the outer solar system
CC	closed cycle
CO	carbon monoxide
CH_4	methane
C_2H_6	ethane
ΔV	change in velocity (km/s)
GCR	gas core rocket
GRC	galactic cosmic rays
GCNR	gas core nuclear rocket
GTOW	Gross Takeoff Weight
H_2	hydrogen
He	helium 4
ISRU	in situ resource utilization
Isp	specific impulse (s)
K	Kelvin
kWe	kilowatts of electric power
LEO	low Earth orbit
M dry, stage	stage dry mass (kg)
M dry, coefficient	stage dry mass coefficient, B

MESSENGER	Mercury Space Surface ENvironment, GEochemistry, and Ranging
MT	metric tons
MWe	megawatt electric (power level)
NEP	Nuclear Electric Propulsion
NPP	Nuclear Pulse Propulsion
NTP	Nuclear Thermal Propulsion
NTR	Nuclear Thermal Rocket
OC	open cycle
O_2	oxygen
PPB	parts per billion

1.0   Introduction

Human and robotic missions have been planned for targets throughout the solar system. Both types of missions can benefit greatly from the resources available from the planets and/or their moons (Refs. 1 to 14). These benefits include water on many of the outer planet moons and large asteroids. With this water, oxygen (O_2)/hydrogen (H_2) rocket propulsion systems can be fueled, breathing O_2 can be extracted, and other life support functions (cooling fluids, etc.) can be facilitated. In addition, the atmospheres of many planets have ready reserves of gases for propellant production. Carbon dioxide on Mars can be separated into O_2 and carbon monoxide (CO) or, with small amounts of hydrogen methane can be produced. The outer planets offer enormous amounts of energetic gases such as hydrogen (H_2), helium 3 (3He), methane (CH_4), ethane (C_2H_6), and ammonia. By using these in-situ resources, robotic precursor missions can double or triple their payloads to the surface and return double or triple the samples from the solar system targets. Without in-situ resource utilization (ISRU), solar system exploration will be exceedingly limited. For future large scale human missions, the possibilities of ISRU for human exploration and settlement offer the best opportunities for sustainability and success.

2.0   Human Exploration Options

In the 1950's, 1960's, 1970's, and 1980's, ambitious robotic and human mission were planned, spanning from Mercury to the outermost reaches of the solar system (Refs. 1 to 14). While investments in robotic missions have continued, human exploration of the solar system has awaited new invigorating steps. While lunar and Mars missions are in the early step-wise planning stages, many cost barriers have prevented their implementation. Future human missions to other destinations such as Mercury and Saturn will also require long-term investments. Currently, Mercury and Saturn have robotic missions returning invaluable data on those planets and their environs (Refs. 15 to 20). These data have provided insights that will ensure the success of future missions. With its proximity to the Sun, Mercury has extremely high temperatures and missions requires special high heat flux considerations for long-term human visits or bases. In contrast, temperatures at Saturn and its moons require designs for cryogenic environments.

2.1   Mercury

Mercury is the closest planet to the Sun; ranging from a perihelion of 46 million km to an aphelion of nearly 70 million km. The high temperature, high heat flux environment at Mercury and the tenuous surface emanations of several major chemical species (sodium, etc.) surrounding it will likely pose challenges to long term human visits. Permanently shadowed craters offer a valuable niche for longer

term human visits and planetary bases. Such craters offer cryogenic temperatures while the sun facing surface is at a temperature of 590 to 725 K. The north polar regions of Mercury have been identified as a likely location for such permanently shadowed craters (Refs. 16 to 18). Water ice is also likely to be in these craters, further aiding and assisting any human exploration. For examples, short exploratory missions could be accomplished with hopping ascent-descent vehicles from a base at the shadowed crater.

Reference 16 shows the locations of the shadowed craters. Reference 17 illustrates the temperatures that would exist in and near the craters: 80 K (in the shadowed area) to 350 K (at the edge of the sunlit region). The craters could accommodate a small base or at least an initial landing site. The lander's temperature could stay within the nominal operating temperatures of traditional spacecraft. The temperature distribution in the crater would allow construction of the base at the warmer side of the crater and then the frozen volatiles would be extracted with cryogenic mining machines. As it may be mixed with regolith and salts, additional purification of the water will be needed.

Additional potentially more complex issues may arise from Mercury's exosphere. The Mariner 10 and MESSENGER spacecraft have measured the planet's exosphere. Figure 1 and Figure 2 (Ref. 21) provide some of the results of these investigations. Through meteoroid vaporization, ion sputtering, and photon-stimulated desorption and thermal evaporation, sodium, calcium, and magnesium atoms are ejected from the surface. The tails of the ejected atoms extend over 33,600 km (21,000 miles) from Mercury. Over the long term, these and other ejected ions may have a deleterious effect on spacecraft surfaces.

Cryogenic cooling and storage of the hydrogen in orbit for the return trip will also be a challenge. Depots using sunshields and specialized radiators may enable this longer term storage. Figure 3 (Ref. 21) provides a description of such cryogenic propellant depots for Earth orbital storage. Figure 4 presents the MESSENGER spacecraft configuration (Ref. 22). Reference 22 describes the cooling system for the MESSENGER spacecraft. A design combining both methods may be effective.

2.2 Saturn and Its Moons

Saturn is one of the outer planets. Its orbit has a perihelion 1,352.6 million km and an aphelion 1,514.50 million km. An extensive series of flybys of the Saturnian moons have been conducted by the Cassini spacecraft. During these flybys, cameras and instruments capture the data on the moons' composition, atmosphere and cloud cover (on the moon Titan), volcanoes, plumes, rotation, and gravity.

Titan is the largest moon of Saturn. Figure 5 shows the possible nature of Titan's interior, surface, and atmosphere (Ref. 18). Its intriguing nature includes a nitrogen and methane atmosphere and a subsurface ocean (Ref. 18). Appendix A presents Titan's atmospheric pressure as a function of altitude (Ref. 20). Recent flybys of the Cassini spacecraft have shown direct visual evidence of the northern lakes. Large lakes in the North Polar regions are likely composed of liquid methane and ethane. Based on gravity measurements and theories of the evolution of Titan, a large ocean of water and ammonia may exist below the icy surface. With purification, the likely water ocean can be an excellent supply of oxygen and hydrogen propellants. While methane and ethane in the lakes and atmosphere can be used as an effective chemical rocket propellants, its nitrogen could be used in cold gas propulsion or electric propulsion (resistojet, arcjet or magneto-plasma-dynamic (MPD) thrusters). In addition, water is flowing out from a subsurface ocean on Enceladus (Ref. 19). Thus, there are several water ice resources on the moons of Saturn.

3.0   Human Space Vehicles and Missions

Space vehicle engine performance, propulsion mass scaling and ΔV estimates were used to predict the low Earth orbit (LEO) masses of both Mercury and Saturn exploration vehicles. LEO mass estimates for extremely high energy missions were assessed.

3.1   Mercury Missions

A human round trip mission to Mercury was assessed. The mission ΔV values for the round trip Mercury missions were derived from the literature (Refs. 23 to 27). The highest ΔV case was selected from this data: an Earth departure ΔV of 5.2 km/s, a Mercury arrival ΔV of 10.9 km/s and a Mercury departure ΔV of 8.7 km/s (Ref. 11). At Earth, a capsule enters the atmosphere to return the crew directly to Earth (Ref. 11). The capsule's mass is 4,350 kg; the round trip time is 585 days with a 40 day stay time at Mercury. In this case, the vehicle does not land on Mercury (Ref. 11). The LEO masses of both chemical propulsion and nuclear thermal propulsion vehicles were estimated. Figure 6 compares the LEO masses for two types of chemical propulsion systems (with the differing tankage mass assumptions noted below) and two nuclear thermal propulsion (NTP) systems. The interplanetary chemical propulsion systems used tankage dry mass coefficients of 3 and 5 percent of the total propellant mass in the tankage. In many cases, these dry masses may be deemed to be optimistically low; however, they allow some relative comparison of the chemical propulsion and the nuclear mission cases.

The NTP vehicles' dry mass was 15 percent of the propellant mass. In current NTP designs, an I_{sp} of 900 sec is nominally used. Somewhat lower I_{sp} values were used for these missions: 800 and 850 sec, respectively (Ref. 11). These lower I_{sp} values were assumed given the high heat flux environment of Mercury and the degraded I_{sp} values would reflect the added propellant used for propellant cooling and/or refrigeration. The chemical propulsion systems required between 17,150 and 31,230 MT to accomplish the mission. The NTP vehicles required approximately an order of magnitude less mass in LEO: 1,700 to 2,300 MT. Based on our prior analysis, the stage and lander mass was estimated with a mass scaling equation (Refs. 23 to 27).

M dry, stage (kg) = M, dry coefficient * M p (kg)

where:

M dry, stage = the stage dry mass, including residual propellant (kg) =

M dry, coefficient = the B mass coefficient (kg of tank mass/kg of usable propellant mass)

M p = usable propellant mass (kg)

A Mercury landing vehicle mass was also estimated; the one-way ΔV for the lander was 3.5 km/s. The ascent ΔV was also 3.5 km/s (Ref. 22). These ΔV values accommodate approximately 19 percent for gravity losses for each maneuver; this gravity loss ΔV is added to the orbital velocity for a 100 km orbit which is 2.945 km/s. The lander I_{sp} was 480 sec. The higher I_{sp} was chosen for the lander as the engine used a higher engine expansion ratio than the interplanetary transfer vehicle. The smaller engine size would allow a higher expansion ratio, given the typical volume constraints for space vehicles. The dry mass coefficient was 20 percent of the total propellant load. While the Mercury missions will likely require more aggressive thermal control (propellant shielding, cooling, etc.), that thermal control system mass is accommodated in the payload mass of the vehicle. The payload delivered to the surface was

10 MT. Figure 7 compares the mass in LEO of a one-way lander and a round trip lander. The masses were 140 MT for the round trip lander and 27 MT for the one-way lander. Thus, using ISRU on the surface of Mercury to replenish the lander's propellant would allow a savings of 113 MT on this mission. Additional analyses are needed to investigate the mass reductions for the interplanetary transfer vehicle to carry the lander to Mercury. Another option would be to carry 5 landers to Mercury rather than carry simply one lander; many more permanently shadowed craters could then be visited on one mission.
The interplanetary vehicle carrying the 5 landers could be sent on a lower energy trajectory than the human flights, thus saving additional mass launched into LEO in the overall Mercury architecture.

Using Mercury resources to augment the human missions was investigated. An ISRU system's effects on reducing the LEO mass (see Table 1 for details of the NTP-1 and NTP-2) was analyzed. For the NTP-1 and NTP-2 systems, cases were computed where the Mercury departure ΔV propellant was supplied at Mercury. The Mercury departure stage is brought from Earth with no propellant. Hydrogen would be produced from the water at the northern polar craters, and transported to orbit. For the NTP-1 case, 94 MT of hydrogen would be transported to orbit. With NTP-2, the propellant mass required in Mercury orbit is 82 MT. With the in-situ hydrogen production, the LEO mass of the NTP-1 case is reduced from 2300 to 760 MT, as shown in Figure 7. Similar large LEO mass reductions are enabled for the NTP-2 system; using ISRU, the 1667 MT LEO mass is reduced to 588 MT.

3.2   Mercury Surface Excursion Planning

While the temperature is quite comfortable for human habitation inside the permanently shadowed craters (PSC), excursions will be desirable to other locations. While the temperature can be quite high outside the PSC, short excursion will be possible. Robotic missions with cooling systems can persist for long periods in the sunlit areas. These explorers can provide data on the most attractive locations for sampling, and the need for human exploration. Cooling systems based on the heat pipe based design from MESSENGER can be effective (Ref. 22).

Hopping out of permanently shadowed craters for short periods will be desirable. Hopper spacecraft have been conceived for many planetary missions (Ref. 28). While the hoppers can be used for excursions outside the PSC, flights into the anti-Sun shadowed regions of the Mercury will allow for more extensive planet wide exploration (or in essence, wait until it is night time).

3.3   Jupiter and Saturn Mission Studies and Results

Human Jupiter and Saturn mission analyses using nuclear pulse propulsion were conducted in the 1960's and 1970's (Refs. 13 to 14 and 29 to 33). Small nuclear devices were detonated behind the vehicle and with a combination of 100's of nuclear devices, a massive pusher plate and shock absorbers, the vehicle is accelerated through the needed ΔV. The LEO masses for varying propulsion dry masses and total mission ΔV assumptions were estimated. These missions used propulsion mass scaling that may have been optimistic. Parametric variation of the dry mass was analyzed.

References 13 to 15 provided the details from which the propulsion system mass scaling equations were derived. The A parameter of 358,000 kg is the mass of the pusher plate and associated shock absorbers for the 20-meter diameter system. The 20-meter diameter system design was used on the outer planet mission analyses conducted in References 14 and 15. The Saturn and the Jupiter mission studies used the same set of propulsion mass scaling equations. The Saturn mission ΔV values were derived from References 13 to 15.

The mass scaling equation was:

$$M_{dry} (kg) = A + B M_p$$

where

M_{dry} = Propulsion system dry mass including residual propellant (kg)

M_p = Usable propellant mass (kg)

A = fixed propulsion system mass (kg)

B = propellant mass dependent mass (kg/kg M_p)

The final set of mass scaling equations were:

$$M_{dry} (kg) = 358,000 + 0.01 M_p$$

Also the parameter B was also varied over a range of values: B = 0.01, 0.02, 0.05, and 0.10.

The NPP space vehicle Isp was 3,000 sec. While a 3,150 sec Isp was used in References 13 and 14, the lower (and more conservative) 3,000 sec value was used in later publications (Ref. 15).

3.4 Jupiter Mission Design and Payload

Jupiter missions were designed to depart LEO, arrive at Jupiter and enter orbit about Callisto (a large Jupiter moon), remain in orbit for 30 to 50 days, depart Jupiter and Callisto and return to LEO. In References 13 to 15 and 29 to 33, the vehicle delivered a payload of 302 MT to Callisto. The crew size was 20 persons and 2 Jupiter moon landers were carried.

Appendix B summarizes the baseline Jupiter mission results. Figure B1 illustrates the LEO masses of several Jupiter missions. Table B1 provides the overall ΔV summary for the 5,920 MT case (for a ΔV of 60 km/s). This LEO mass was one of the highest masses of all Jupiter mission NPP vehicle designs. Table B2 summarizes many other Jupiter mission designs. Lower ΔV cases were also conducted and the lowest ΔV of 20 km/s represented a LEO mass of nearly 1,500 MT. A series of low and high ΔV missions are summarized in Figures B2 and B3.

3.5 Saturn Missions

Figures 9 through 16 provide the mass estimates for a series of Saturn missions over a range of mission ΔV. Based on the work of References 13 to 14, and 29, a range of mission ΔV from 60 to 120 km/s was used in the calculations. While the highest ΔV values may represent impractically high LEO masses for some space missions, they are presented for comparison and completeness. The same overall mission payload masses were used for human Jupiter and Saturn missions.

Figure 9 to Figure 16 show the variations in LEO mass for B coefficients of 0.01, 0.02, 0.05, and 0.1. The variation in B coefficient would reflect the variation in the mass of the feed systems required for nuclear pulse unit storage, transfer, and ejection. Reference 29 noted that the B coefficient would likely be 0.01. However, this mass could easily increase given the complexity of the feed systems and the need for multiple canisters to store the individual nuclear units. Each canister was designed to hold 100's of nuclear units.

For vehicles delivering a ΔV of 60 km/s, the LEO masses were 6,000 MT for the case of B = 0.01 to 7600 MT for a B = 0.10. In many cases, space vehicles experience mass growth during development. As such, a more conservative mass estimate is always good for comparison. The vehicle masses for the highest ΔV cases are over 48,000 MT to 97,000 T. These LEO masses are unusually high and will require many specialized heavy lift launch vehicles. Reference 33 noted the development of the NEXUS vehicle, a post Saturn V capability of over 450 MT (nearly 1 million lbm) into LEO. Reference 33 noted the need for a NEXUS launch capability of 4 to 8 times that of the Saturn V (implying up to nearly 1,000 MT into orbit). Certainly, high Isp options using fusion propulsion for the interplanetary vehicle would simplify operations and reduce the number of launches to LEO.

Even with an increase launch capability, the number of 1,000 MT payload launches required would likely be higher than 50. This number would also include cryogenic boiloff makeup (for propulsion, life support, science support, etc.), airborne support equipment, and other contingency masses.

4.0   Observations

Saturn mission data was derived from several historical sources, detailing ambitious human missions. While on-orbit assembly of these more massive vehicles (of over 10,000 MT) is not impossible, it would seem likely that other higher Isp propulsion options would be more attractive. Nuclear electric with hundreds of megawatts or nuclear fusion designs could deliver Isp values of 10,000 to 30,000 sec (Ref. 34) and have the potential to significantly reduce the LEO mass over the NPP options.

5.0   Concluding Remarks

Propulsion systems for human missions to Mercury and Saturn were assessed. While there are many thermal challenges with the Mercury environment, the potential of permanently shadowed craters in the northern polar region can provide a safe haven with cool temperatures for human landings and habitation. Nuclear thermal propulsion (NTP) without ISRU can enable round trip human Mercury missions for a LEO mass of 1,700 to 2,300 MT. Using hydrogen produced on Mercury, an ISRU mission can reduce that LEO mass to 590 to 760 MT, respectively. Landers using only Earth based oxygen/hydrogen can land 10 MT on the surface and return 10 MT to orbit. With Mercury ISRU and by refueling the landers on Mercury' surface, 5 landers can be delivered to Mercury on one NTP mission, each carrying 10 MT on round trip flights.

Saturn missions may require a LEO mass of at least 6,000 MT for a mission ΔV of 60 km/s. Missions with up to 120 km/s ΔV were analyzed. While LEO masses of 10,000 to 100,000 MT and more are needed for such missions, larger future space operations may be able to accommodate such massive vehicles. Due to the relative frailty of the human body (subject to deep space radiation and lower gravity), fast missions to the outer planets will be required. Ambitious space missions using higher energy propulsion will be needed to enable such high speeds. Investments in higher energy nuclear thermal propulsion would be the first best step. Incremental investments in more advanced propulsion system will benefit not only human and robotic missions, but many benefits will be gleaned in numerous other technological areas: space based power production, expanded planetary locations for human population growth, etc. These technological innovations will enable Krafft Ehricke's vision of a poly-global civilization, which will indeed benefit all of humanity and all of its endeavors.
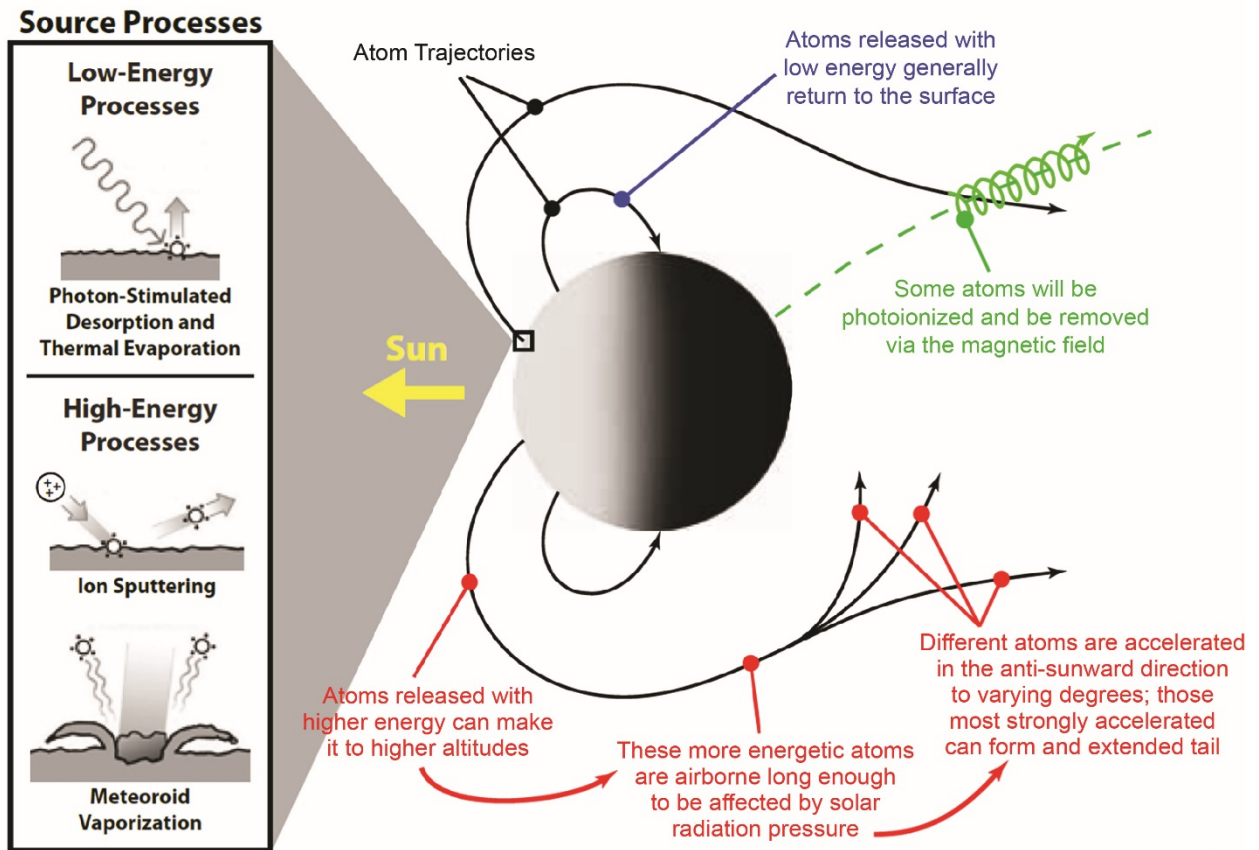
Processes at Work in Mercury's Exosphere
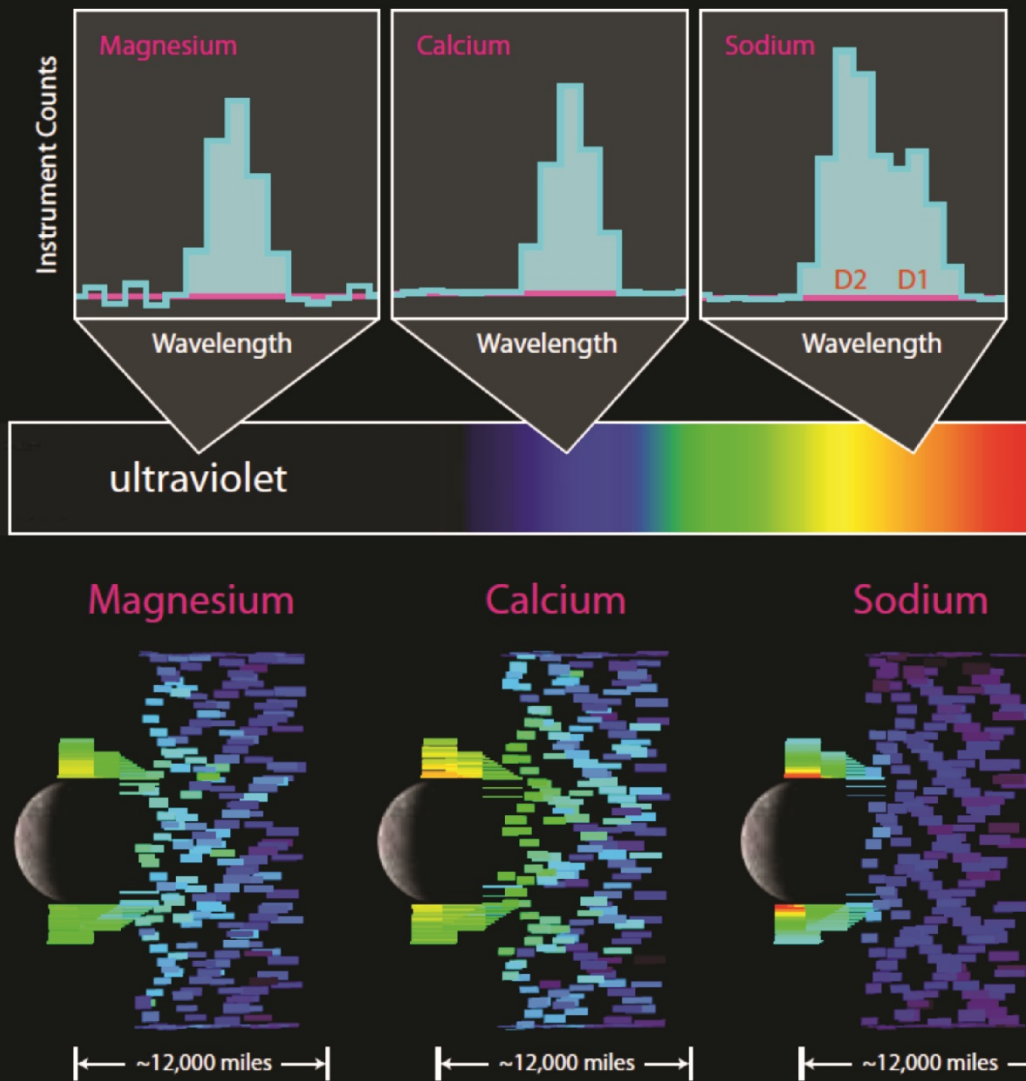

Figure 1.—Processes at work in Mercury's Exosphere (Ref. 21).

Figure 2.—Elements emanating from Mercury due to solar flux (Ref. 21).

Simple Depot Extesibility to Larger Depots

5 m LH$_2$ tank enables 70 MT depot capacity
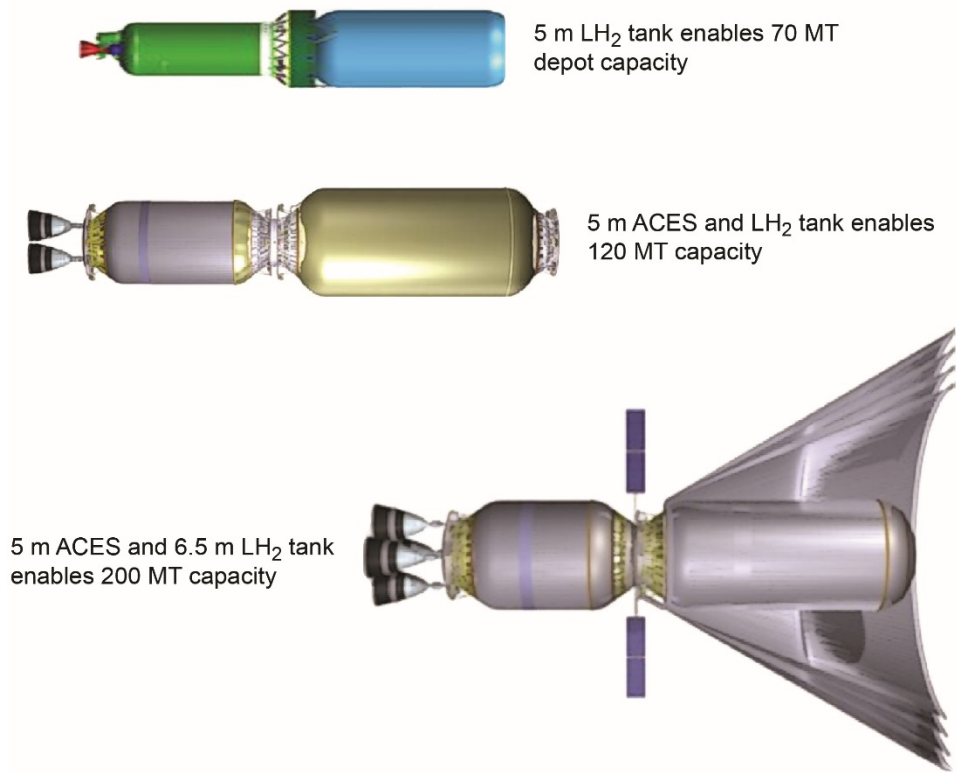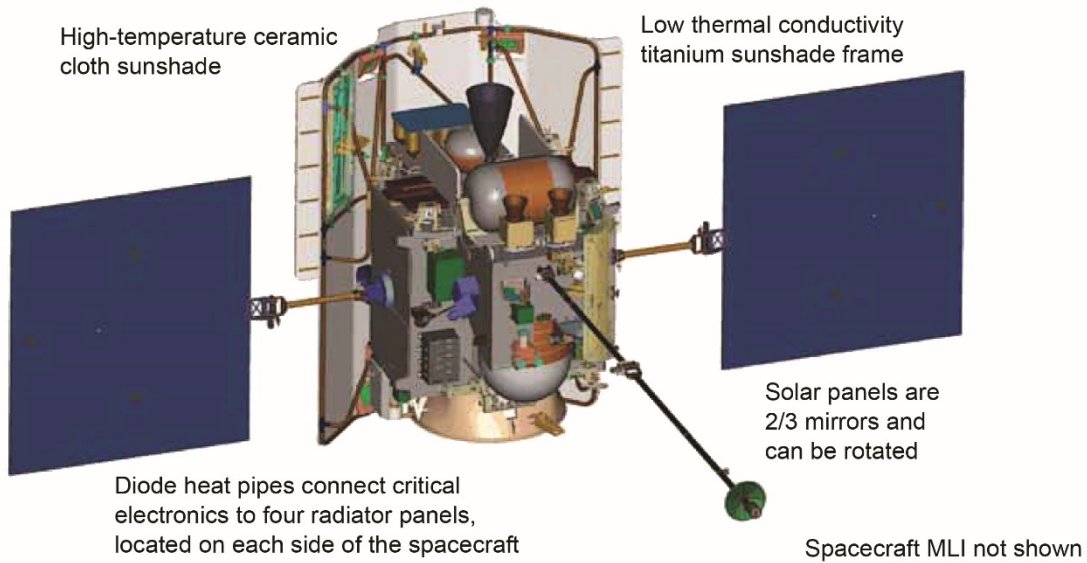
5 m ACES and LH$_2$ tank enables 120 MT capacity

5 m ACES and 6.5 m LH$_2$ tank enables 200 MT capacity

Figure 3.—Cryogenic propellant depot options (Ref. 20), "Cryogenic Propellant Depots Design Concepts and Risk Reduction Activities," Future In-Space Operations (FISO), teleconference, *March 2, 2011,* Christopher McLean (Ball Aerospace).

High-temperature ceramic cloth sunshade

Low thermal conductivity titanium sunshade frame

Solar panels are 2/3 mirrors and can be rotated

Diode heat pipes connect critical electronics to four radiator panels, located on each side of the spacecraft

Spacecraft MLI not shown

Messenger spacecraft in-flight configuration, rear view

Figure 4.—MESSENGER spacecraft configuration (Ref. 21).
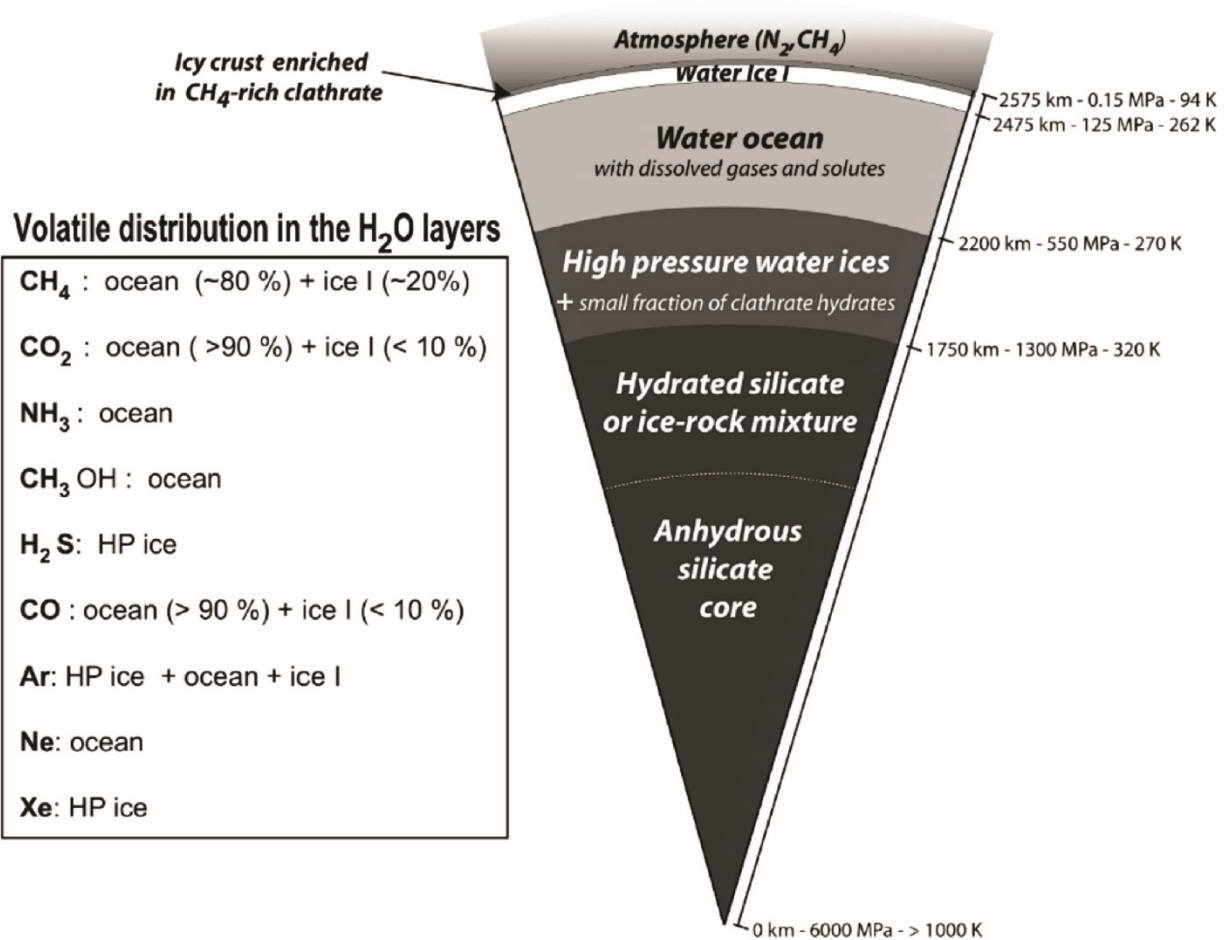
Figure 5.—Possible present day cross section of Titan (Ref. 18, reprinted with the permission of Cambridge University Press.).

Human Mercury Mission, Total Mission ΔV = 24.8 km/s

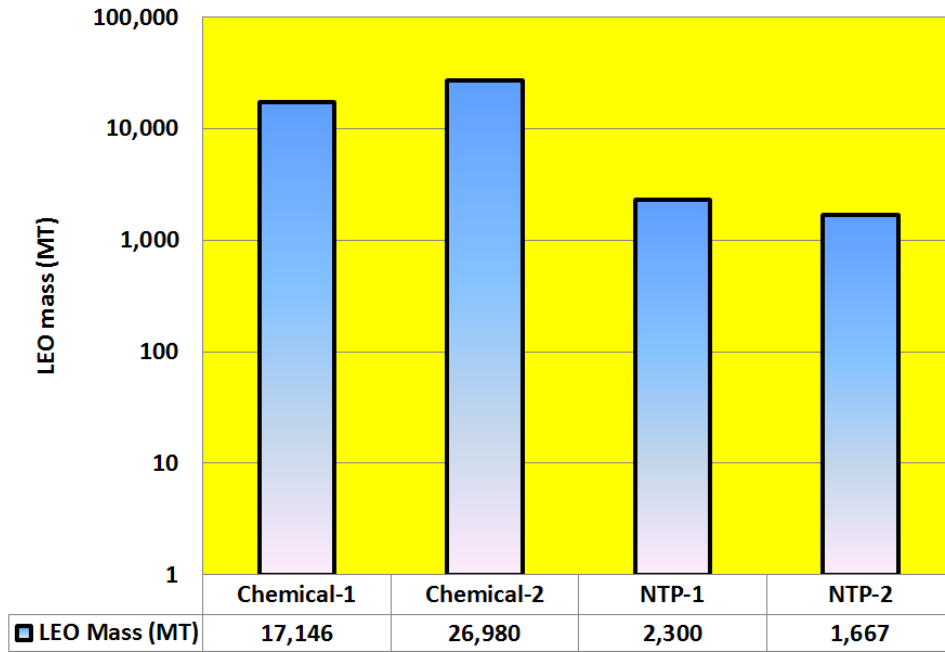|	Chemical-1	Chemical-2	NTP-1	NTP-2
☐ LEO Mass (MT)	17,146	26,980	2,300	1,667

Figure 6.—LEO Departure Mass for Human Mercury Missions.

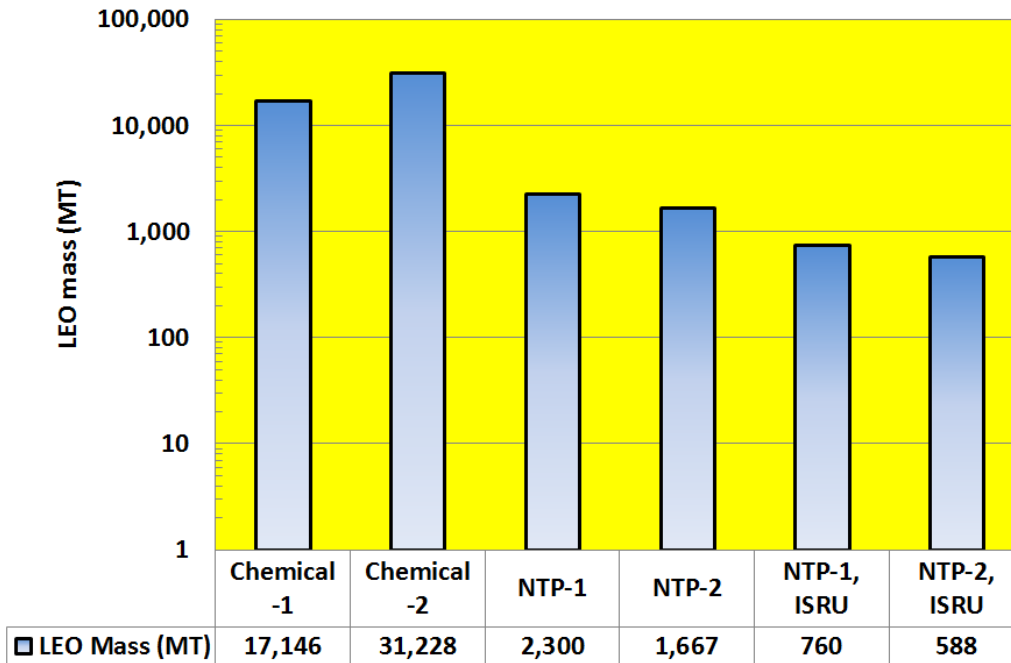**Human Mercury Missions, total mission, ΔV = 24.8 km/s, 8.7 km/s Mercury departure ΔV (with NTP ISRU options)**

	Chemical-1	Chemical-2	NTP-1	NTP-2	NTP-1, ISRU	NTP-2, ISRU
☐ LEO Mass (MT)	17,146	31,228	2,300	1,667	760	588

Figure 7.—LEO masses of human round trip missions to Mercury (without and with ISRU).

**Lander Mass, Chemical Propulsion, Payload = 10 MT,
ΔV = 3.5 km/s each, for Descent and Ascent**

	Chemical lander (2-way)	Chemical lander (1-way)
☐ LEO Mass (MT)	140.1	27.0
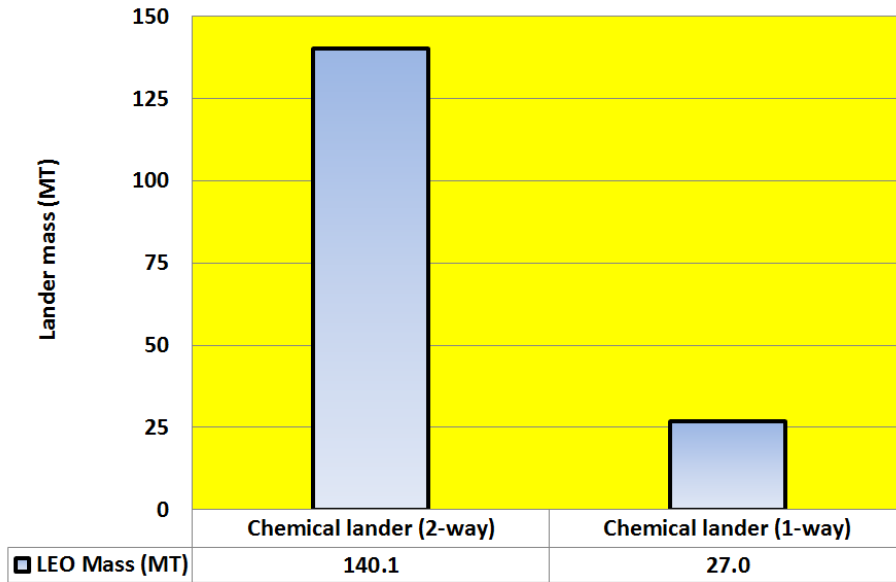
Figure 8.—Mercury Lander Mass (2-way (without ISRU) and 1-way (ISRU)).

**Human Saturn Missions, NPP, four maneuvers,
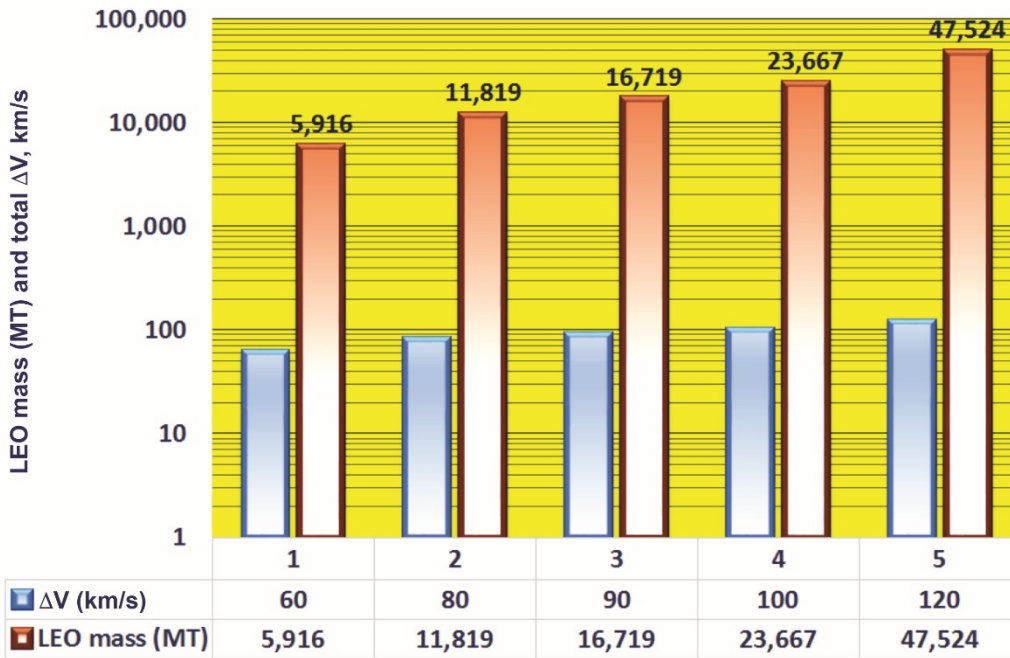Isp = 3,000 sec 358 MT pusher plate, B = 0.01**

	1	2	3	4	5
☐ ΔV (km/s)	60	80	90	100	120
☐ LEO mass (MT)	5,916	11,819	16,719	23,667	47,524

Figure 9.—Saturn mission data, B = 0.01 Mp.
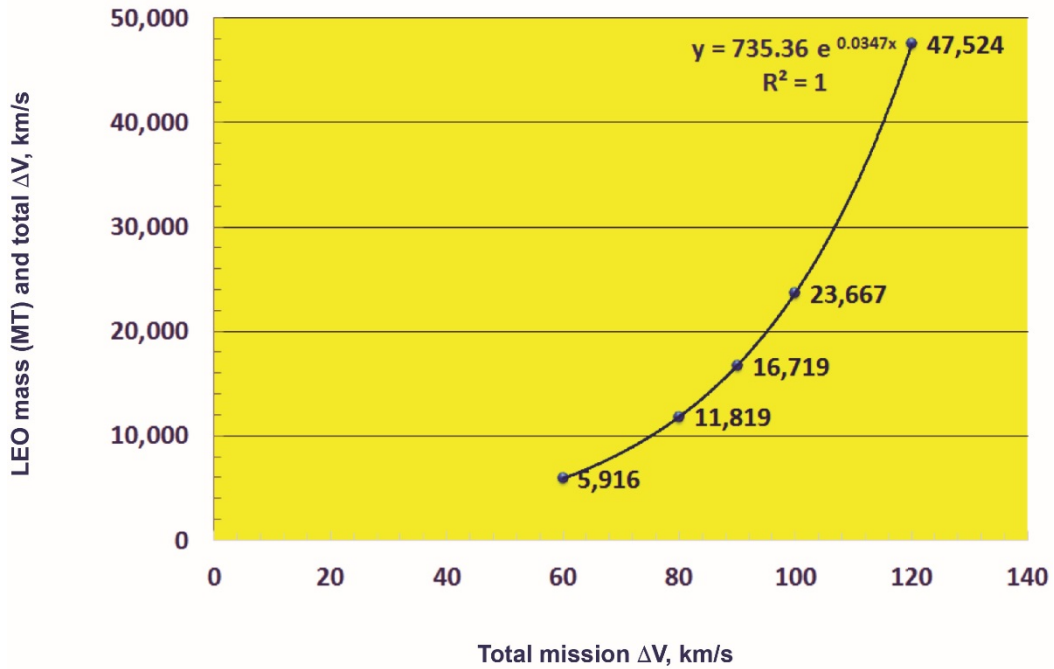
Figure 10.—Saturn mission data, correlation, B = 0.01 Mp.
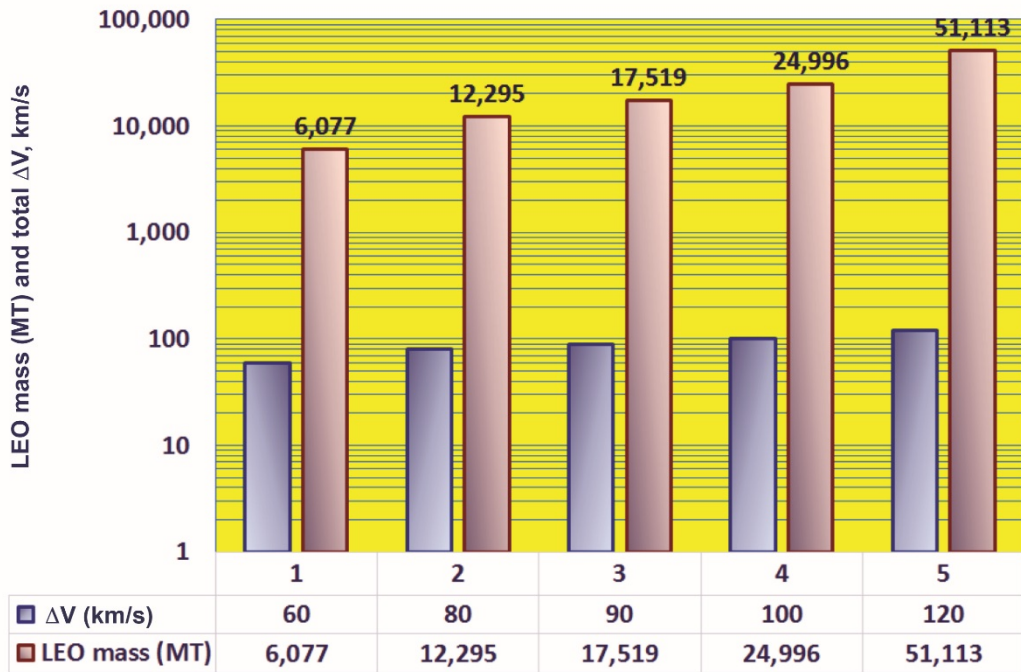
Figure 11.—Saturn mission data, B = 0.02 Mp.
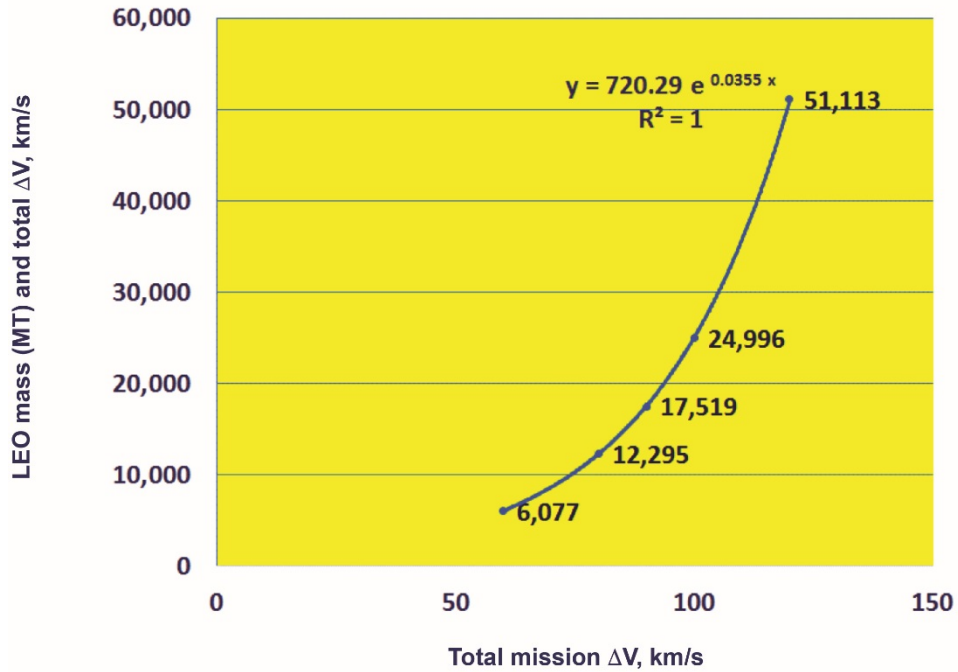
Figure 12.—Saturn mission data, correlation, B = 0.02 Mp.
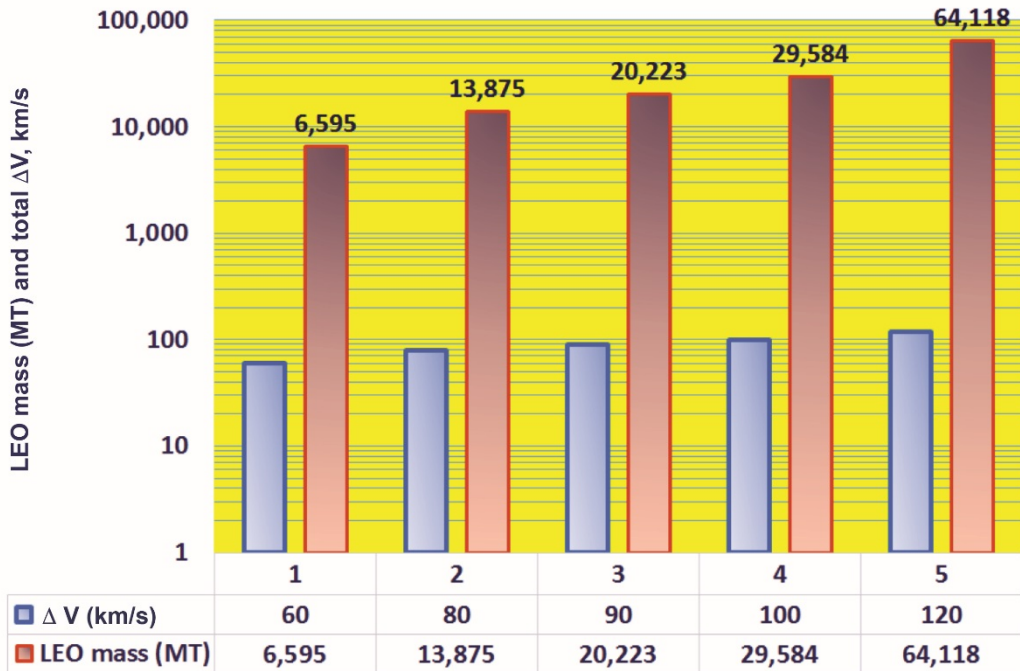
Figure 13.—Saturn mission data, B = 0.05 Mp.
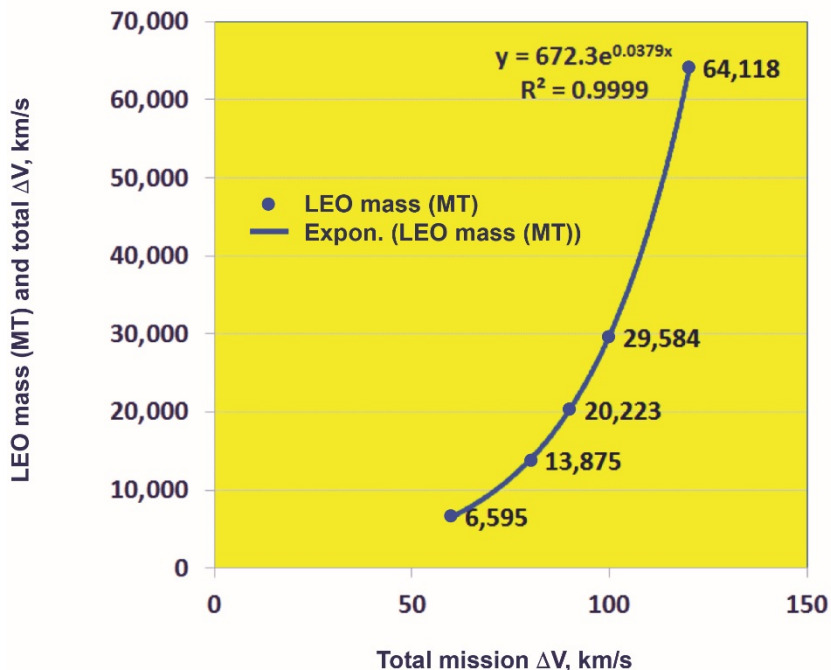
Figure 14.—Saturn mission data, correlation, B = 0.05 M.
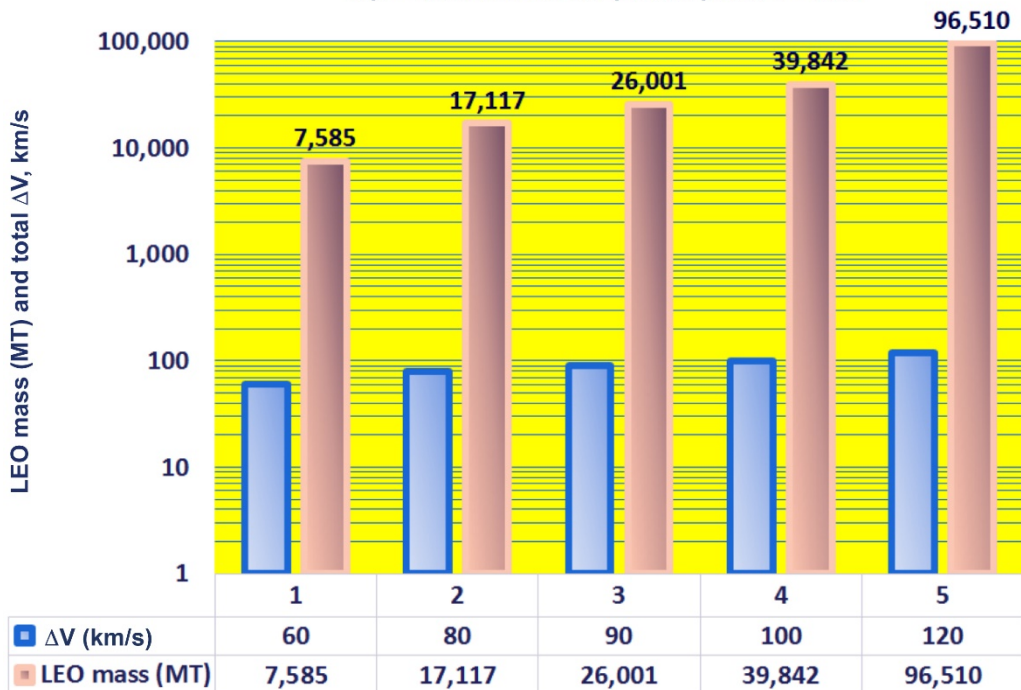
Figure 15.—Saturn mission data, B = 0.10 Mp.
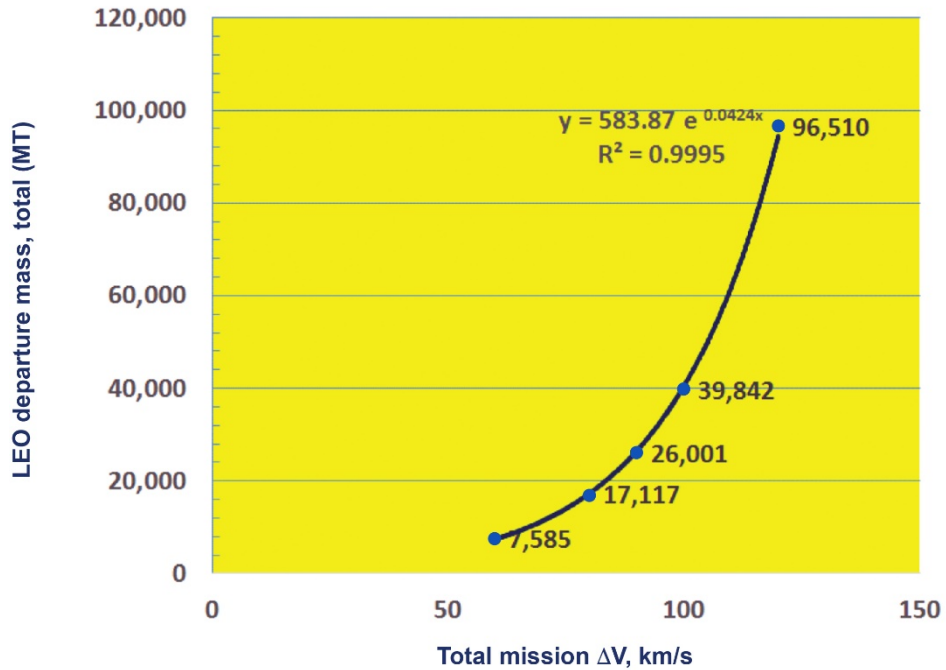
Figure 16.—Saturn mission data, correlation, B = 0.10 Mp.

TABLE 1.—SPACE VEHICLE DRY MASS COEFFICIENT AND
ROCKET ENGINE SPECIFIC IMPULSE (I_{sp})

Technology	I_{sp} (sec)	M, dry coefficient (kg/kg M,p)
Chemical-1	450	0.03
Chemical-2	450	0.05
Chemical lander	480	0.20
NTP-1	800	0.15
NTP-2	850	0.15

Appendix A.—Titan's Atmospheric Temperature, Pressure, and Density
(Ref. 22)

Chart created from data in: Fulchignoni, M., et al. "In situ measurements of the physical characteristics of Titan's environment," Nature 438, 785-791, 8 December, 2005 (Figure A1).
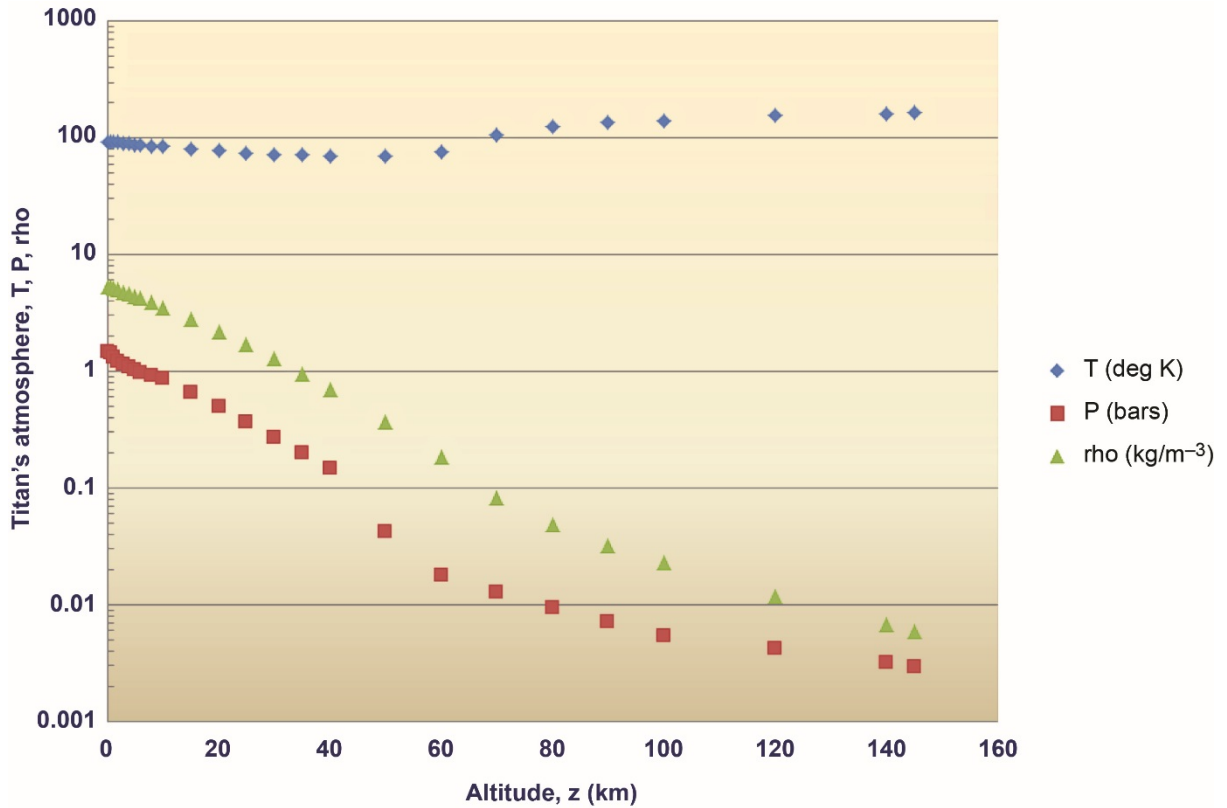
Figure A1.—Titan's atmospheric pressure versus altitude.

Appendix B.—Jupiter Mission Analyses and Data (Ref. 29)

TABLE B1.—JUPITER MISSION DATA (REF. 29)

Maneuver	ΔV requirement	
	m/sec	ft/sec
Earth departure	25,000	81,800
Outbound correction	610	2,000
Callisto capture	12,300	40,400
Callisto departure	8,900	29,200
Return correction	610	2,000
Earth arrival	16,320	53,600
Total velocity required	63,740	209,000
	Dates and trip time, days	
Reference departure date	2/20/82	
Outbound transfer time	400	
Capture and window times	30, 20	
Return transfer time	460	
Mission duration	910	

Human Jupiter Missions, NPP, four maneuvers, Isp = 3,000 sec 358 MT pusher plate, B = 0.01

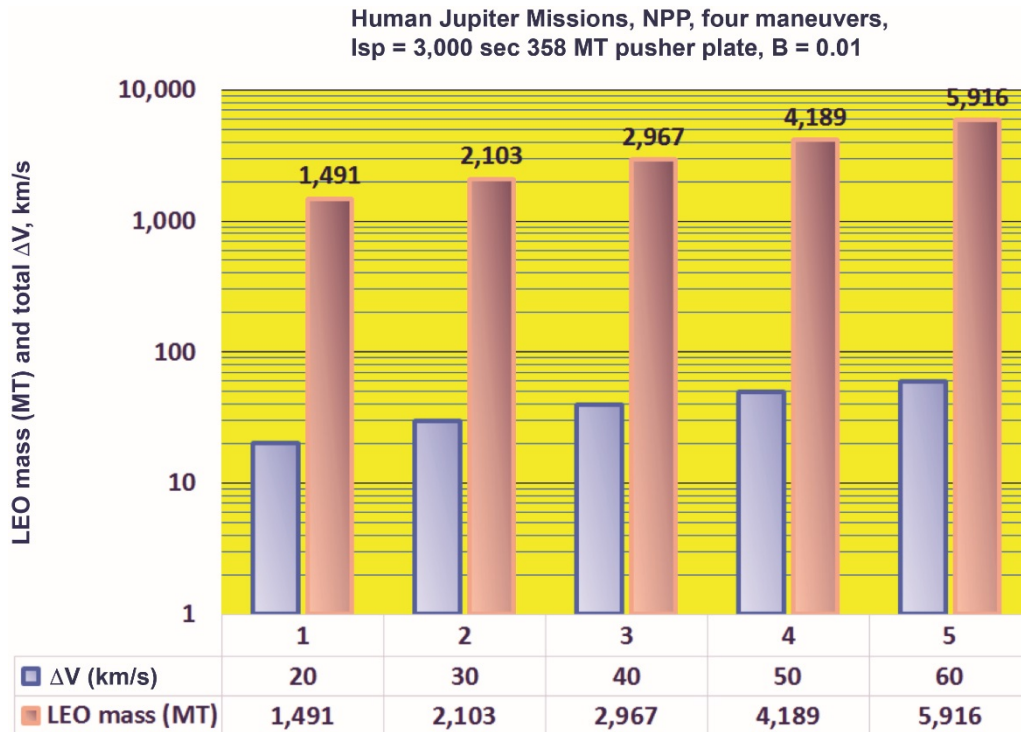|	1	2	3	4	5
ΔV (km/s)	20	30	40	50	60
LEO mass (MT)	1,491	2,103	2,967	4,189	5,916

Figure B1.—Jupiter mission data.

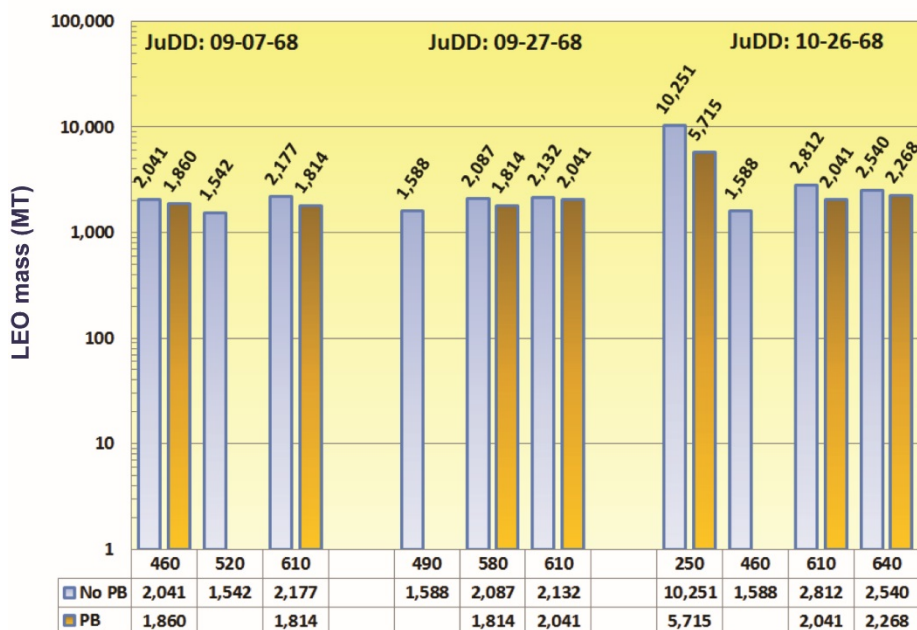## Human Jupiter Missions, Round Trip Time: NPP 3,000 sec Isp

Figure B2.—Human Jupiter missions using NPP (data from Ref. 29).

TABLE B2.—HUMAN JUPITER MISSIONS USING NPP (REF. 29)

Planet	EaDD	T1 (days)	V∞1 (EMOS)	ΔV2 (EMOS)	V∞2 (EMOS)	ΔV2 (ft/sec) n=3, r* = 1.1	T cpt (days)	JuDD	T2 (days)	V∞3 (ft/sec)	ΔV3 (EMOS)	V∞4 (EMOS)	Ve (10³ ft/sec)
Ju	12/3-12/21, 1980	460	----	28,000	----	33,000 10,000 (n = 30)	----	----	----				
Ju	12/29, 1981-1/22, 1982	460	0.4	27,300	0.53	33,000 10,000 (n = 30)	----	----	----				
Ju	2/3-2/17, 1983	460	0.38	24,700	0.513	32,000 9,600 (n = 30)	----	----	----				
Ju	3/8-3/21, 1984	460	0.37	24,400	0.492	32,000 9,600 (n = 30)	----	----	----				
Ju	4/10-4/28, 1985	460	0.37	24,400	0.472	31,500 8,600 (n = 30)	----	----	----				
Ju	5/18-6/6, 1986	460	0.37	24,400	0.456	30,000 8,200 (n = 30)	40	10/15, 1987	520	0.39	29,500 6,900 (n = 30)	0.40	54,000
Ju	6/24-7/9, 1987	460	0.36	24,100	0.452	31,000 8,200 (n = 30)	72	12/27, 1988	460	0.462	31,200 8,200 (n = 30)	0.40	54,000
Ju	8/2-8/17, 1988	460	0.38	24,700	0.46	31,200 8,300 (n = 30)	79	1/28, 1990	460	0.478	31,500 8,700 (n = 30)	0.40	54,000
Ju	9/6-9/21, 1989	460	0.39	25,000	0.48	33,000 8,800 (n = 30)	79	----	460	0.498	32,000 9,200 (n = 30)	0.40	54,000

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